

# NAVAL RESEARCH LABORATORY NAVAL CENTER FOR SPACE TECHNOLOGY

Design Loads and Analysis Requirements  
for the  
Full-Sky Astrometric Mapping Explorer (FAME)

**NCST-D-FM017A**

**27 November 2001**

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## 1. INTRODUCTION

### 1.1 Background

The Full-sky Astrometric Mapping Explorer (FAME) flight vehicle consists of a spacecraft (SC) bus, interstage assembly, and a single primary instrument. The instrument will provide the positions, proper motions, parallaxes, and photometry of nearly all stars as faint as 15<sup>th</sup> visual magnitude with an astrometric accuracy between 50 to 500 microarcseconds, dependent on the magnitude. The instrument is supplied by Lockheed-Martin Space Systems (LMSS), Palo Alto, CA.

The bus and interstage assembly are built by NRL. NRL is responsible for integrating the instrument to the bus.

### 1.2 Purpose

This plan establishes and defines the tasks associated with the structural design, loads, analysis, and analysis verification for the FAME flight vehicle. It is the intent of this document to describe the activities, methods and responsibilities of the structural design and analysis groups associated with this program.

The plan addresses the use of a Delta II 7425-10 launch vehicle for establishment of the design environments and criteria. A Star 30BP is assumed for the apogee kick motor (AKM).

### 1.3 Scope

The requirements of this document shall apply to all FAME flight hardware and ground support equipment.

It is to be applied to vendor supplied items including the instrument and its components.

### 1.4 Definitions

Interstage Assembly: The Interstage assembly provides the structural load path between the launch vehicle and the spacecraft. It also serves to support the Star 30BP AKM.

The lower interface is at the Marmon clamp attachment to the launch vehicle (LV). The clamp band is considered part of the LV since it remains with the LV after separation.

The upper interface is the Marmon clamp interface with the spacecraft. The clamp band itself is included since it remains with the interstage assembly after separation.

SC Bus Assembly: The NRL supplied SC bus includes the primary spacecraft structure and bus mounted subsystems, such as propulsion and power.

The lower interface is the Marmon clamp interface with the interstage assembly. The clamp band itself is not included since it remains with the interstage assembly after separation.

The upper interface is located at the lower mounting surfaces of the instrument bi-pod flexure mounts.

FAME Flight Vehicle: Same as the LV payload. Includes the interstage assembly, SC bus, and instrument.

Instrument Assembly: The instrument is the primary source of science data. It is made by LMSS. It includes the instrument itself, bi-pod flexure mounts, and NRL supplied star trackers and omni antenna.

The lower interface is formed by the plane of the mounting surfaces of the flexures.

Spacecraft: The orbiting segment of the FAME flight vehicle. This includes the SC bus and the instrument.

Component: From the standpoint of mechanical analysis and testing, components are functional items that are viewed as a complete and separate entity for the purposes of manufacturing, maintenance, or record keeping. An example of such components are all the electrical boxes and instrument mirrors.

Subsystem: From the standpoint of mechanical analysis and testing, subsystems are assemblies of functional related components. They may consist of two or more components with interconnection items. An example of such subsystems are the instrument assembly.

System: From the standpoint of mechanical analysis and testing, systems are an integrated set of subsystems capable of supporting an operational role. An example of such systems are the SC and flight vehicle assemblies.

Primary Structure: The primary structure is defined as all structural components that make up the primary load paths for the SC and flight vehicle structural components.

Secondary Structure: The secondary structure is defined as all structural components that are not part of the primary load path. In general, these components include mounting brackets, etc.



## **2. APPLICABLE DOCUMENTS**

### **2.1 Standards, Specifications and Program Documents**

<b>Number</b>	<b>Title</b>
MIL-STD-1540D	Product Verification Requirements for Launch, Upper Stage, and Space Vehicles
MIL-HDBK-340A	Test Requirements for Launch, Upper Stage, And Space Vehicles Vol 1: Baselines
MIL-HDBK-340A	Test Requirements for Launch, Upper Stage, And Space Vehicles Vol II: Applications Guidelines
MIL-HDBK-5H	Metallic Materials and Elements for Aerospace Vehicle Structures
NASA-TM-X-73305	Astronautic Structures Manual, Vol. I
NASA-TM-X-73306	Astronautic Structures Manual, Vol. II
NASA-TM-X-73307	Astronautic Structures Manual, Vol. III
MSFC-SPEC-522B	Design Criteria for Controlling Stress Corrosion Cracking
MSC/NASTRAN	Users Manual, Vol. I & II
MSC/NASTRAN	Application Manual, Vol. I & II
MIL-HDBK-17E	Composites Materials Handbook
MIL-STD-1522A	Standard General Requirements for Safe Design and Operation of Pressurized Missile and Space Systems
NCST-ICD-FM001	FAME Spacecraft Bus to Instrument Interface Control Document (ICD)
NCST-D-FM002	FAME Mission Requirements Document (MRD)
NCST-TP-FM001	FAME System Integration and Test Plan
SSD-PS-018	Process Specification Installation of Bolts, Screws, Washers, Nuts and Hi-Lok Fasteners
MDC-00H0016 (Oct 2000)	Delta II Payload Planner's Guide

### **2.2 Reports**

Not applicable.

### **2.3 Books**

<b>Author</b>	<b>Title</b>	<b>Publisher</b>
Bruhn, E.F.	Analysis and Design of Flight Vehicle Structures	Tri-State Offset Company, 1973
Peery, D.J.	Aircraft Structures	McGraw-Hill, 1950
Young, W.C.	Roark's Formulas for Stress and Strain	McGraw-Hill, 6 <sup>th</sup> edition, 1989

### **2.4 Software**

- Unigraphics computer aided design software package
- MSC/NASTRAN finite element program
- SDRC/IDEAS finite element model pre and post-processor
- FEMAP finite element model pre and post-processor
- MATLAB numeric computation software package
- MATHCAD calculation software for performing and presenting hand analysis

### **3. DESIGN REQUIREMENTS**

#### **3.1 Design Criteria.**

The FAME flight vehicle system shall be designed to survive the launch and ascent, operational, ground handling, and protoflight load environments without permanent deformation and fit into the dynamic envelop of the launch vehicle. In addition, the FAME flight vehicle system shall have sufficient stiffness to exceed the launch vehicle control frequencies.

#### **3.2 Materials**

##### **3.2.1 Material Selection and Properties**

Metallic materials shall be selected from MSFC-SPEC-522, Table 1. Materials other than Table 1 may only be used with approval. Material properties shall be taken from MIL-HDBK-5 or other acceptable sources. A-Basis properties shall be used for metallic materials.

##### **3.2.2 Structural and Metallic Materials**

Use requirements as defined in FAME Mission Requirements Document (MRD), NCST-D-FM002.

##### **3.2.3 Composite Materials**

Use requirements as defined in FAME Mission Requirements Document (MRD), NCST-D-FM002.

##### **3.2.4 Stress Corrosion**

Use requirements as defined in FAME Mission Requirements Document (MRD), NCST-D-FM002.

##### **3.2.5 Material Outgassing**

Use requirements as defined in FAME Mission Requirements Document (MRD), NCST-D-FM002.

##### **3.2.6 Magnetic Materials**

Use requirements as defined in FAME Mission Requirements Document (MRD), NCST-D-FM002.

##### **3.2.7 Finishes**

Use requirements as defined in FAME Mission Requirements Document (MRD), NCST-D-FM002.

#### **3.3 Minimum Frequency**

The stiffness of the FAME flight vehicle system shall produce fundamental frequencies above 35 Hz in the thrust axis and 20 Hz in the lateral axis for the system hard-mounted at the launch vehicle separation plane per launch vehicle requirements. The stiffness of all major components; subsystems, instrument, etc., shall produce fundamental frequencies above 50 Hz in all three axes hard-mounted at their interface.

#### **3.4 Hardware**

##### **3.4.1 Structural Fasteners**

Structural fasteners, other than rivets, shall be made from A286 Stainless Steel, Titanium, or other suitable high strength materials. Fasteners made from 300 series CRES Stainless Steel shall not be used in any primary structural applications.

##### **3.4.2 Fastener Preload**

Fasteners under NRL responsibility will be torqued per SSD-PS-018, unless otherwise specified, to give required preloads. There will be no gapping at ultimate loads.

### 3.4.3 Sandwich Structures

Structural components utilizing sandwich construction shall be design according to the following requirements:

1. The sandwich facings shall be at least thick enough to withstand chosen design stresses under design loads.
2. The core shall be thick enough and have sufficient shear rigidity and strength so that overall sandwich buckling, excessive deflection and shear failure will not occur under design loads.
3. The core shall have high enough moduli of elasticity, and the sandwich shall have great enough flatwise tensile and compressive strength so that wrinkling of either facing will not occur under design loads.
4. The cell size of the honeycomb core shall be small enough so that dimpling of either facing into the core spaces will not occur under design loads.

The choice of materials and methods of sandwich assembly used for design shall be compatible with the expected environment in which the sandwich is to be utilized.

### 3.5 Worst Case Design Masses and Cgs

The FAME flight vehicle system shall be designed to handle the loads generated from the quasi-static design limit loads as stated using the design masses. These masses are intended for stress analyses only and are not intended for maximum mass growth.

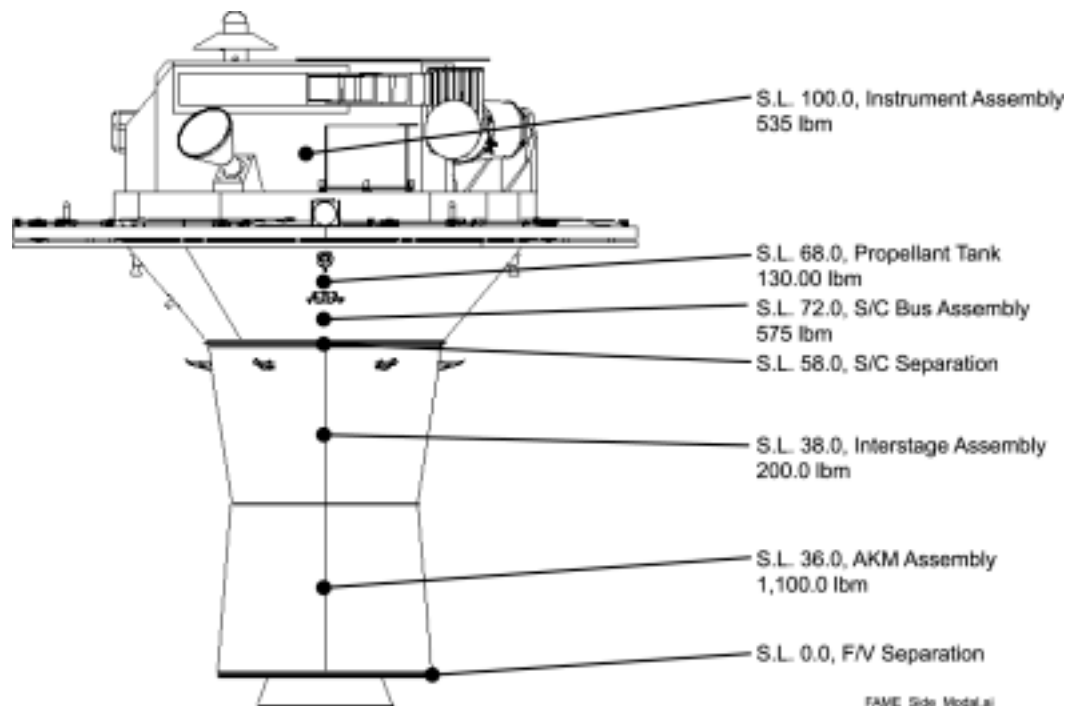


Figure 3-1. Maximum Design Masses

## 4. LOAD REQUIREMENTS

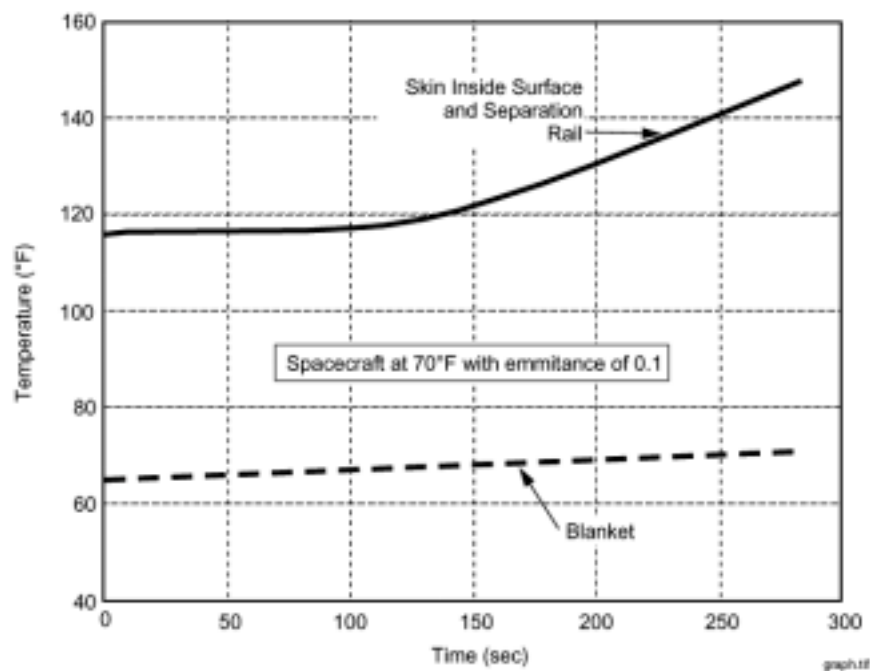
### 4.1 Load Criteria

The FAME flight hardware shall be tested to protoflight test levels in accordance to FAME System Integration and Test Plan, SSD-TP-FM001. The protoflight test approach is intended to combine verification of adequate design margin and adequacy of spacecraft manufacturing and workmanship by subjecting the flight spacecraft to protoflight test levels. In general, the flight hardware shall be designed for proof or protoflight testing to 1.05\*Design Limit Loads and/or flight levels +3 dB for random vibration and acoustics.

### 4.2 Temperature Design Limits

#### 4.2.1 Ascent Thermal Environment

The thermal environment for the FAME S/C is given in Figure 4-1. These data represent the thermal environment for a Delta II 7425-10 Launch Vehicle.



**Figure 4-1. Predicted Maximum Internal Wall Temperature and Internal Surface Emittance (10-ft Fairing)**

#### 4.2.2 Spacecraft Structure Launch Thermal Conditions

The spacecraft will be maintained at a temperature of 69 to 80 degrees F immediately prior to launch. During the ascent phase the structure will not exceed 95 degrees F.

#### 4.2.3 Spacecraft On-orbit Temperature Distribution

The spacecraft on-orbit temperature distributions will be calculated for various orientations and orbital parameters. The temperatures will be used for the calculation on thermal distortions of the spacecraft.

FAME components, subsystems, and systems shall be designed for testing at the test temperature limits specified in Table 4-1. The number of temperature cycles components undergo at ambient pressure and/or vacuum shall be as specified in NCST-TP-FM001, FAME System Integration and Test Plan.

**Table 4-1. Temperature Design Limits**

Type	Test Temperature Limits
Acceptance	5°C above and below the predicted temperature range
Protoflight	10°C above and below the predicted temperature range
Qualification	15°C above and below the predicted temperature range

### 4.3 Launch Pressure Design Limits

The pressure environment for the FAME S/C is given in Figure 4-2. These data represent the pressure environment for a Delta II 7425-10 Launch Vehicle.

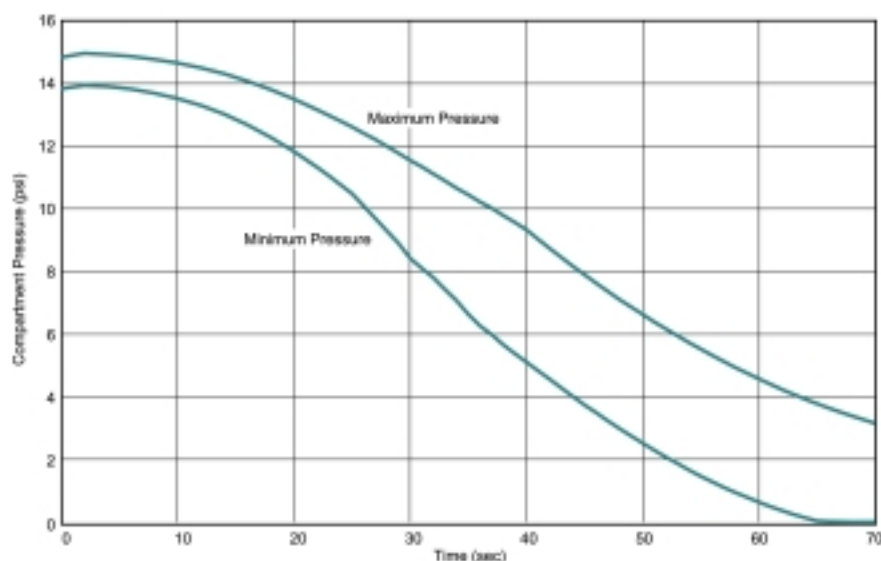


Figure 4-2. Delta II Payload Fairing Compartment Absolute Pressure Envelope

**Figure 4-2. Pressure Profile and Depressurization Rates for Delta II 10-ft dia. Fairing**

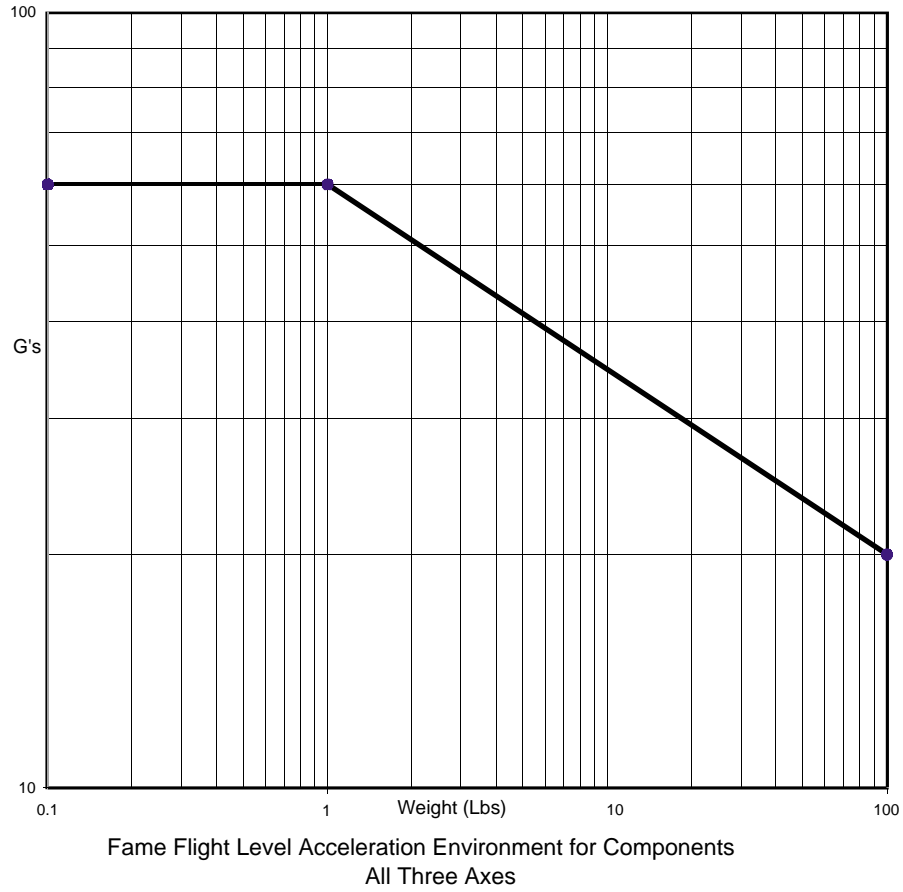
### 4.4 Static Acceleration Design Limit Loads

#### 4.4.1 Component

The mass/acceleration curve (MAC) of Figure 4-3 provides a simplified approach for generating component static loads. Given the weight of a component, the design acceleration can be obtained from the curve. This approach is intended for use with compact components such as electronic boxes, sensors, and other equipment that exhibit high modal frequencies and low sensitivity to mounting flexibility. It is also applied to the design of the component to spacecraft attachment system, component support bracketry, harness restraint, RCS tubing, and heat pipe attachments.

This curve is to be used in addition to the other design criteria for the component. For instance, an electronics box is also designed to meet random vibration test and venting requirements.

The MAC shall be applied in three critical orthogonal axes, one axis at a time.



Design Accelerations		Design Acceleration Philosophy	
Component Wt. (Lbs)	G's	* These accelerations are to be used for component testing by sine burst or centrifuge,  - Appropriate factors of safety shall be applied to these accelerations  - For designated components, the acceleration level from this curve may also be used for vibration test tailoring	
0.1	60		
1	60		
100	20		

**Figure 4-3. FAME Flight Level Mass-Acceleration Curve for Components, All Three Axes**

#### 4.4.2 FAME System, Instrument, and Propellant Tank

Systems, subsystems and their mounts shall be designed to the static accelerations in Table 4-2:

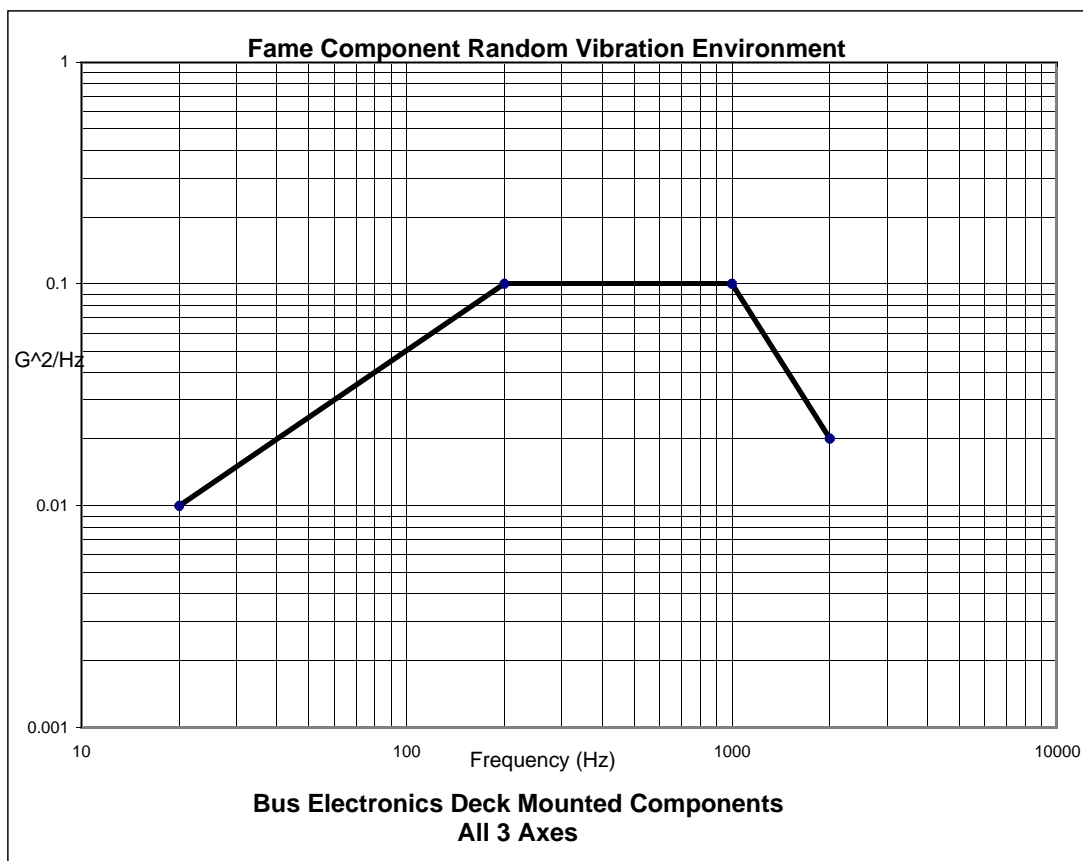
**Table 4-2. Quasi-Static Design Limit Loads**

LOAD CASE	FAME System (1)			Instrument and Propellant Tank (2)			
	Axial + = I/F compression g	Lateral any direction g	Spin	Axial + = I/F compression g	Lateral any direction g	Pitch + Yaw Rotational Accelerations rad/s <sup>2</sup>	Spin
LV Liftoff	+2.8 -0.2	+/- 3.5	0	+2.8 -0.2	+/- 10.8 TBR	+/- 28.9 TBR	0
LV MECO	+7.0	+/- 0.2	0	+7.0	+/- 0.5		0
LV Third Stage Spin Up	0.0	0.0	8.0 rad/sec <sup>2</sup> TBR				8.0 rad/sec <sup>2</sup> TBR
TECO	+5.60	+/- 0.1	60 rpm TBR	+5.60	+/- 0.1		60 rpm TBR
SC AKM Firing	+6.25	+/- 0.1	60 rpm	+6.25	+/- 0.1		60 rpm
Handling/Transport	+/- 2.0	+/- 2.0	0	+/- 2.0	+/- 2.0		0
Notes:							
(1) Lateral, axial, and spin accelerations are applied simultaneously.							
(2) Lateral, axial, pitch, and yaw rotational and spin accelerations are applied simultaneously.							

#### 4.5 Random Vibration Design Limit Loads

##### 4.5.1 Component

The flight level random vibration environments for components are specified according to the component mounting location and orientation, either normal or lateral to the mounting plane. The random vibration environments in the normal and lateral directions for both bus and instrument mounted components, mirrors, and propellant tank are given in Figure 4-4 through Figure 4-11. The random vibration environment for the overall FAME system and instrument alone is given in Figure 4-12. The protoflight test level for the flight unit is +3 dB higher than the flight level for a duration of 1 minute and the qualification test level for the engineering model is +6 dB higher than flight level, also for a duration of 2 minutes.

**Flight Level Environment**

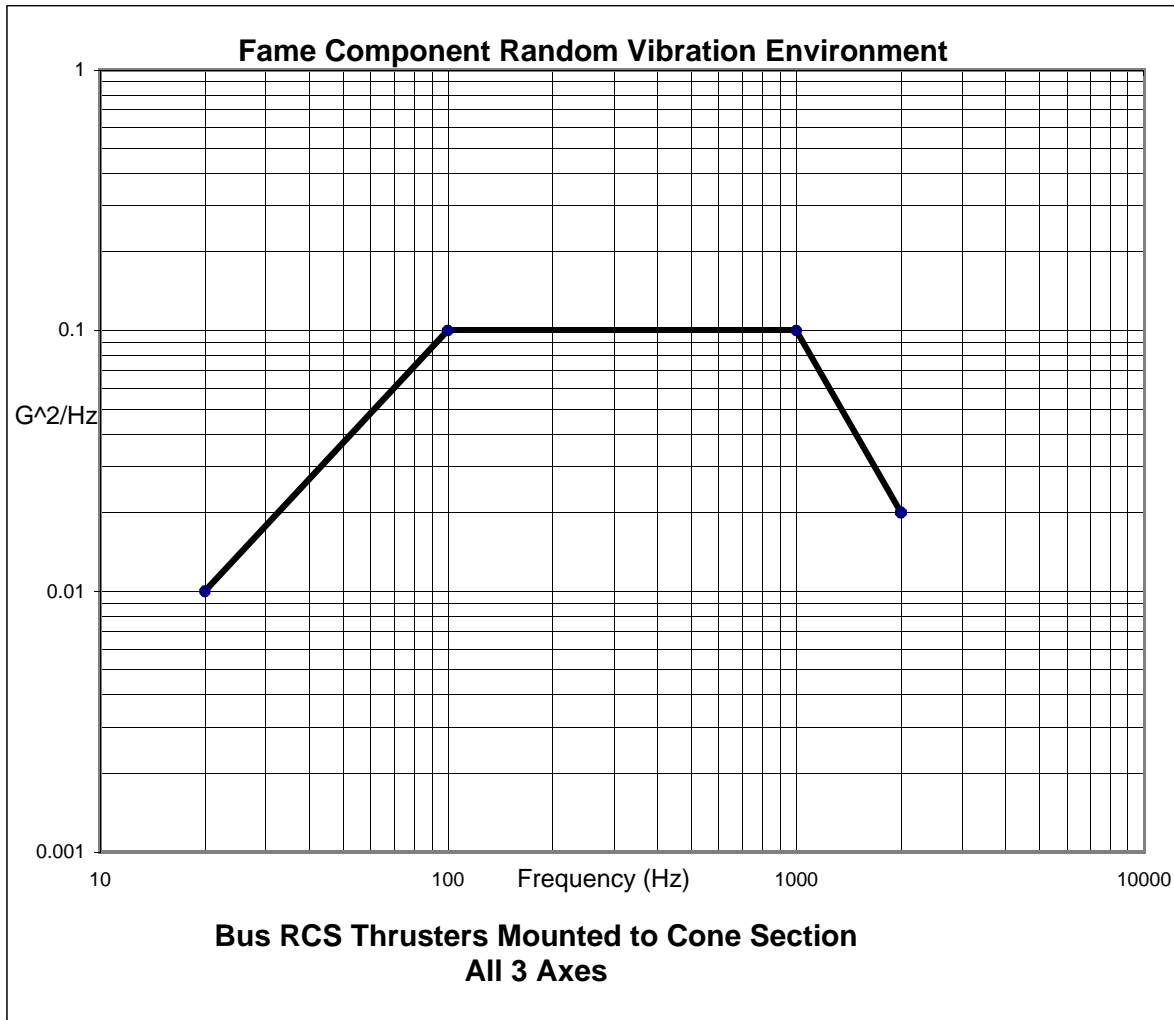
Frequency (Hz)	G <sup>2</sup> /Hz
20	0.01
200	0.1
1000	0.1
2000	0.02
11.6 Grms	

**Test Levels**

	Margin Above Flight Level (dB)	Duration (Minutes)
Non-Flight Prototypes (Design & Qualification Level )	6	2
Flight Units ( Protoflight Acceptance Test)	3	1

**Figure 4-4. Random Vibration Environment, Bus Mounted Components on Electronics Deck,  
All Three Axes**





**Flight Level Environment**

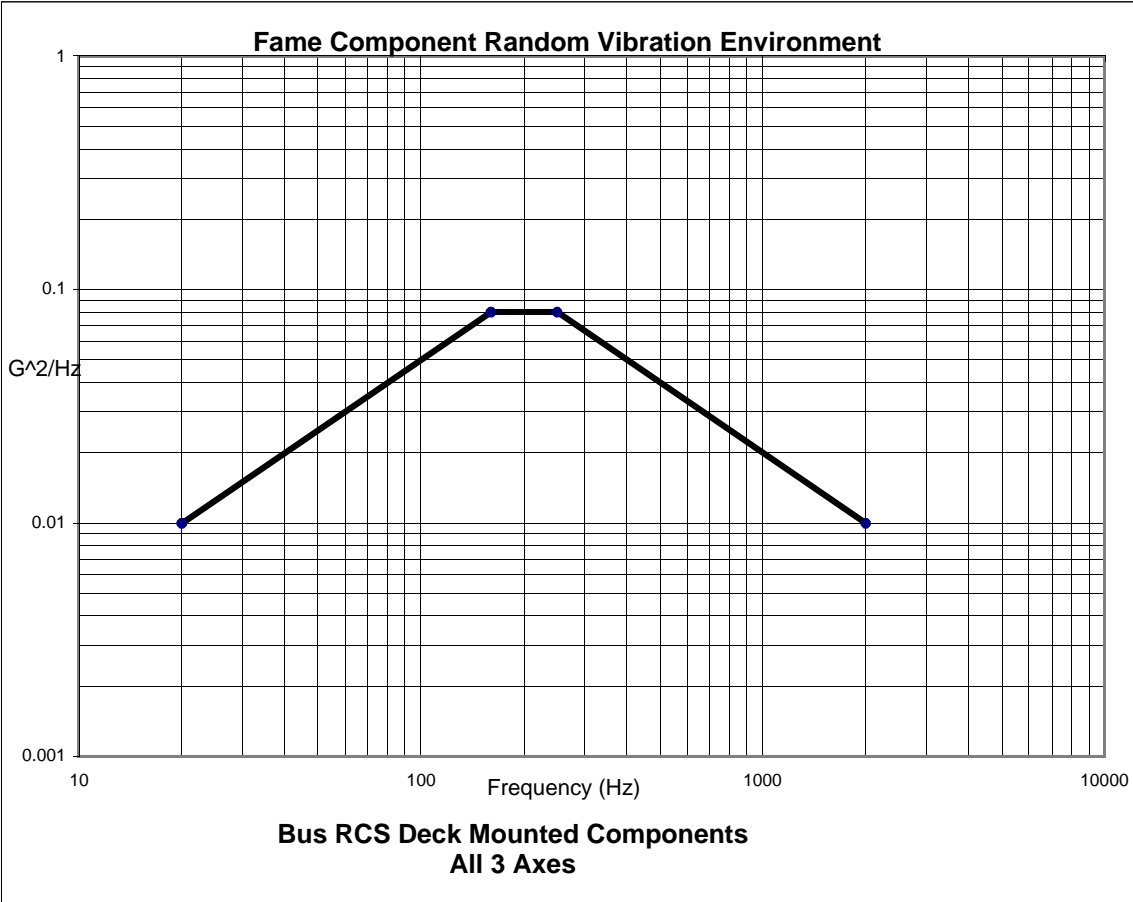
Frequency (Hz)	G <sup>2</sup> /Hz
20	0.01
100	0.1
1000	0.1
2000	0.02

11.8 Grms

**Test Levels**

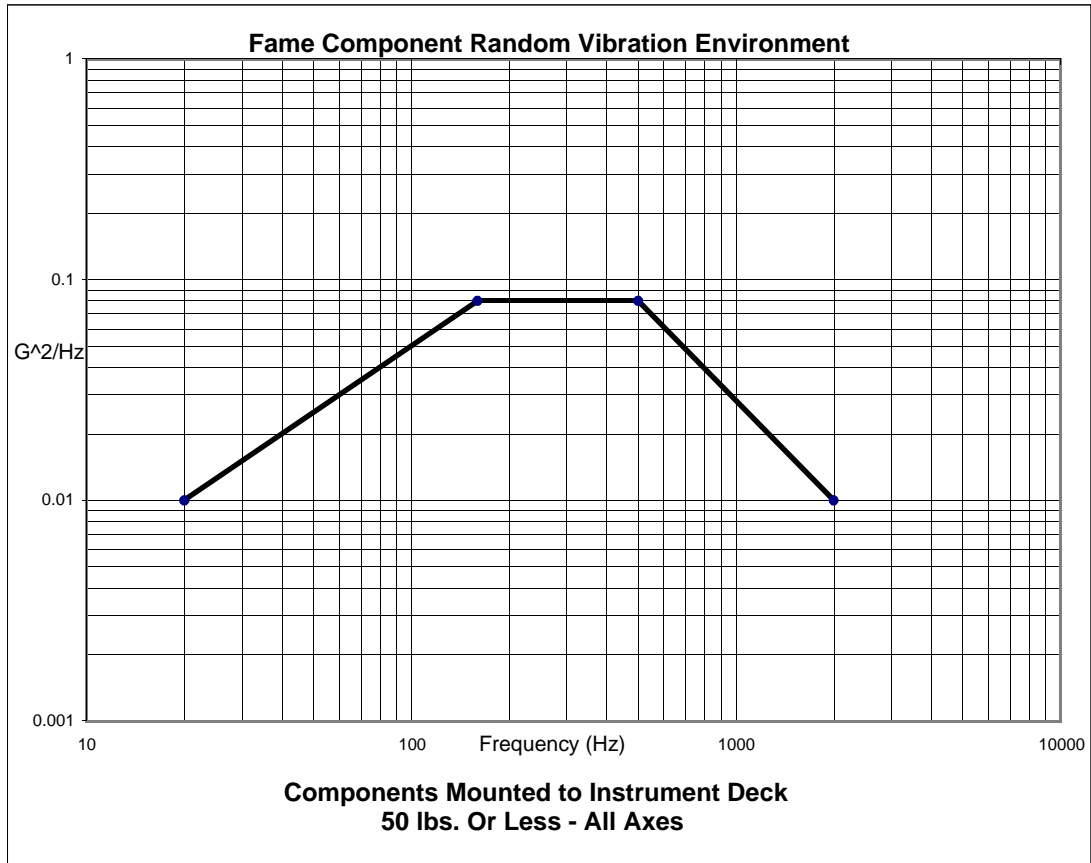
	Margin Above Flight Level (dB)	Duration (Minutes)
Non-Flight Prototypes (Design & Qualification Level )	6	2
Flight Units ( Protoflight Acceptance Test)	3	1

**Figure 4-5. Random Vibration Environment, Bus Mounted Components on Thrust Cone,  
All Three Axes**



Flight Level Environment		Test Levels	
Frequency (Hz)	G^2/Hz	Margin Above Flight Level (dB)	Duration (Minutes)
20	0.01	6	2
160	0.08		
250	0.08		
2000	0.01	3	1
7.4 Grms			
		Non-Flight Prototypes (Design & Qualification Level )	
		Flight Units ( Protoflight Acceptance Test)	

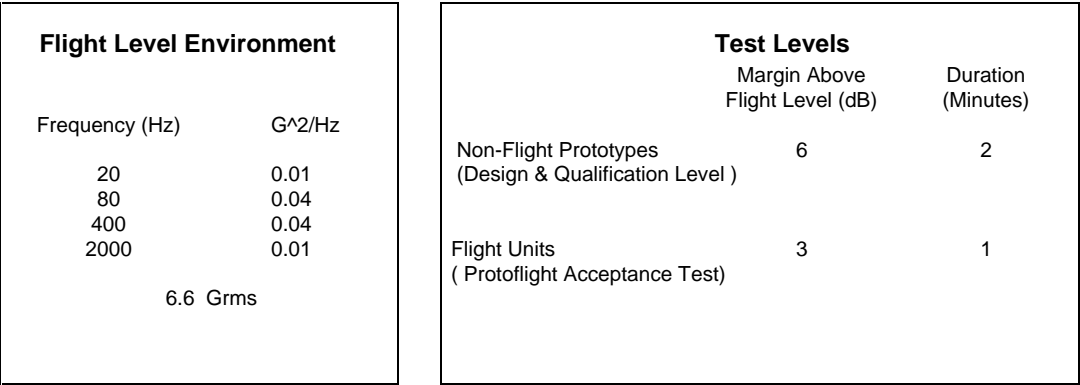
**Figure 4-6. Random Vibration Environment, Bus Mounted Components on RCS Deck, All Three Axes**

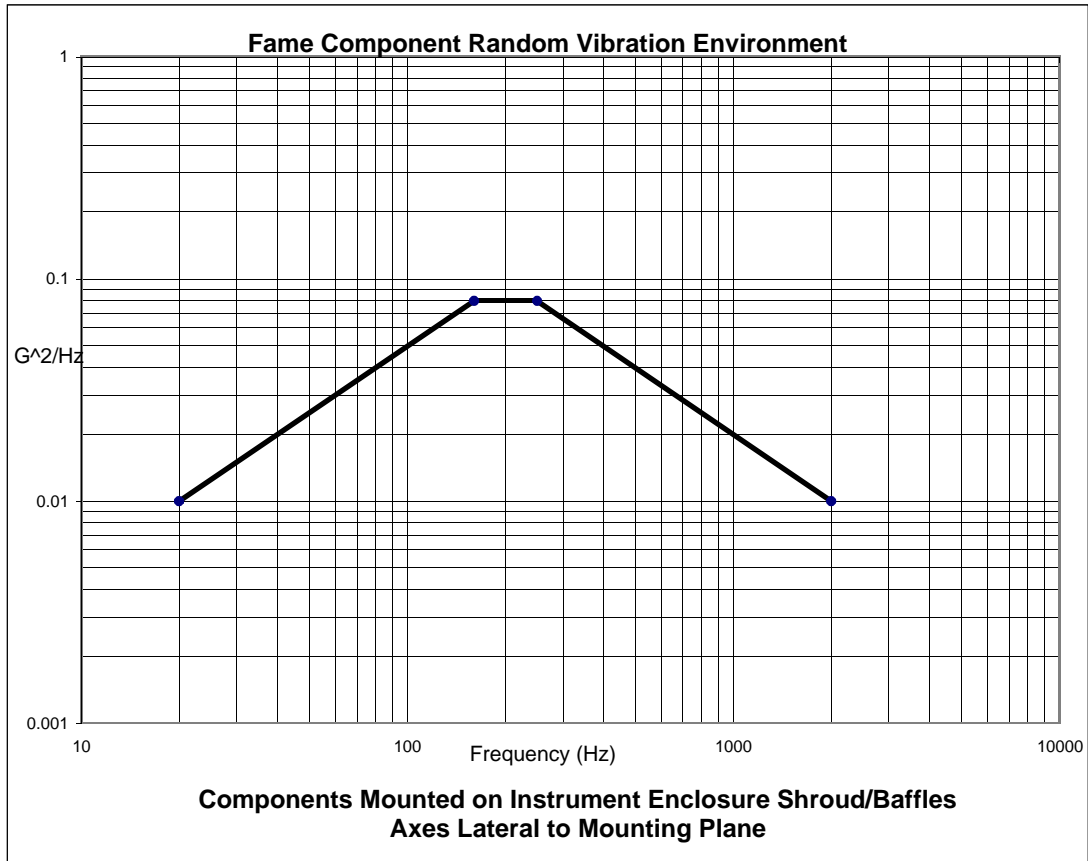


Flight Level Environment	
Frequency (Hz)	G <sup>2</sup> /Hz
20	0.01
160	0.08
500	0.08
2000	0.01
8.6 Grms	

Test Levels		
	Margin Above Flight Level (dB)	Duration (Minutes)
Non-Flight Prototypes (Design & Qualification Level )	6	2
Flight Units ( Protoflight Acceptance Test)	3	1

**Figure 4-7. Random Vibration Environment, Instrument Deck Mounted Components Less Than 50 lb, All 3 Axes**

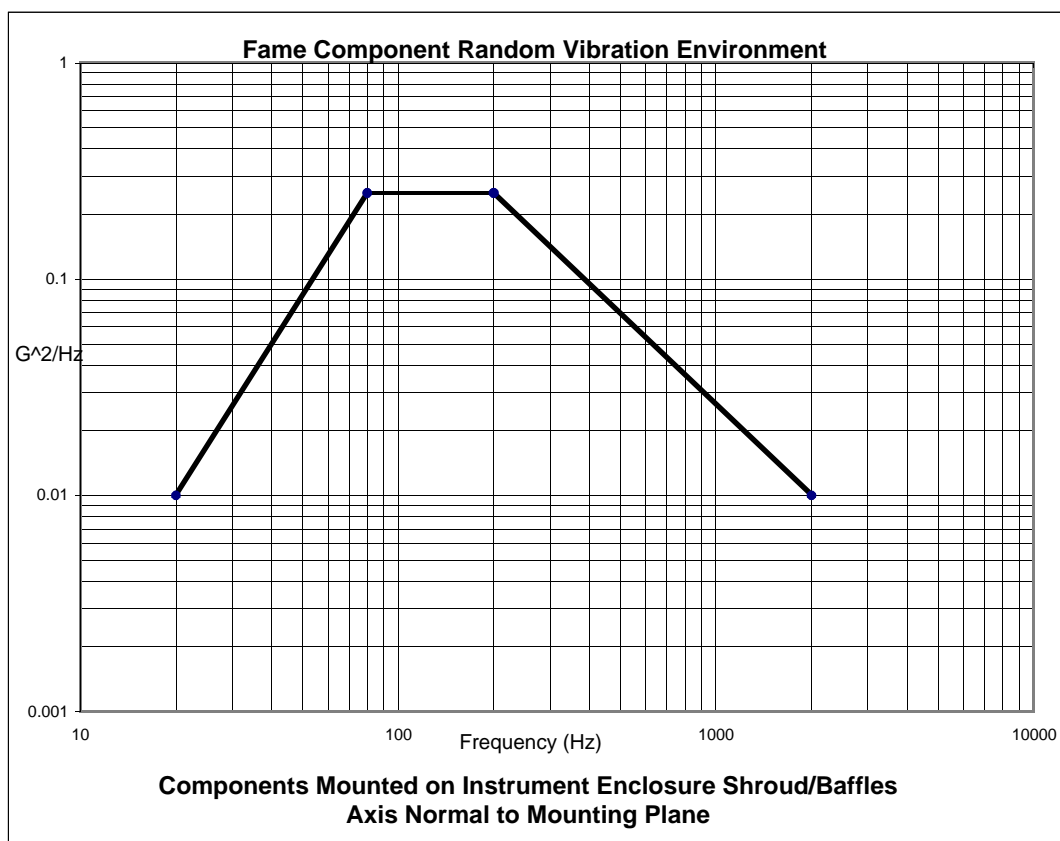




Flight Level Environment	
Frequency (Hz)	G <sup>2</sup> /Hz
20	0.01
160	0.08
250	0.08
2000	0.01
7.4 Grms	

Test Levels		
	Margin Above Flight Level (dB)	Duration (Minutes)
Non-Flight Prototypes (Design & Qualification Level )	6	2
Flight Units ( Protoflight Acceptance Test)	3	1

**Figure 4-9. Random Vibration Environment, Instrument Enclosure Shroud/Baffle Mounted Components, Axis Lateral to Mounting Plane**



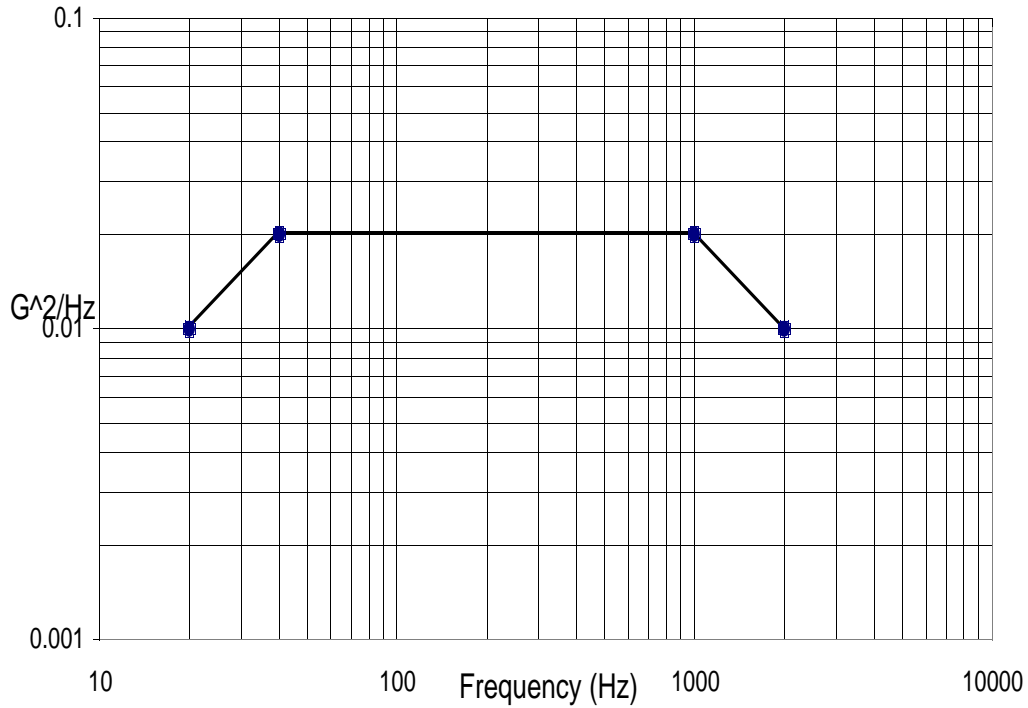
<b>Flight Level Environment</b>	
Frequency (Hz)	G <sup>2</sup> /Hz
20	0.01
80	0.25
200	0.25
2000	0.01
10.6 Grms	

<b>Test Levels</b>		
	Margin Above Flight Level (dB)	Duration (Minutes)
Non-Flight Prototypes (Design & Qualification Level )	6	2
Flight Units ( Protoflight Acceptance Test)	3	1

**Figure 4-10. Random Vibration Environment, Instrument Enclosure Shroud/Baffle Mounted Components, Axes Normal to Mounting Plane**

#### 4.5.2 Propellant Tank

The flight level random vibration environment for the empty propellant tank is given in Figure 4-11. The protoflight test level for the flight unit is +3 dB higher than the flight level for a duration of 2 minutes and the qualification test level for the engineering model is +6 dB higher than flight for a duration of 2 minutes.



Flt Level Environment	
20	0.01
40	0.02
1000	0.02
2000	0.01
5.8 Grms	

Test Level		
	Margin Above Flt Level (dB)	Duration (Minutes)
Non-Flt Prototypes (Design & Qual Level)	6	2
Flight Units (Protoflt Accept Test)	3	2

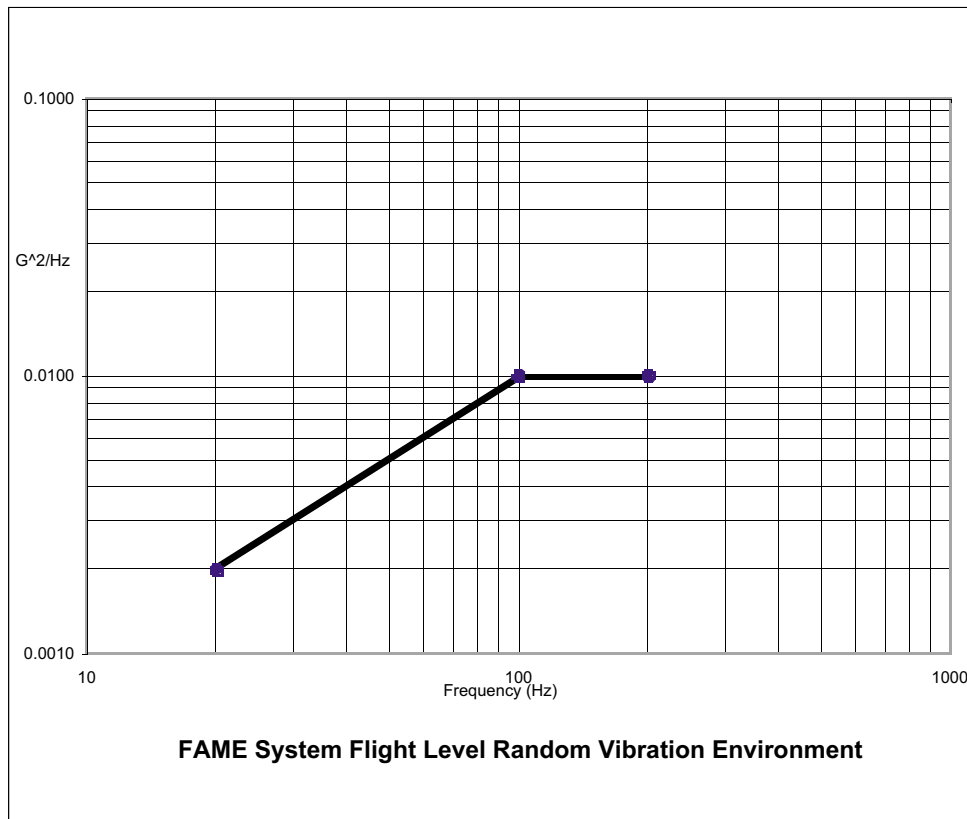
Figure 4-11. Empty Propellant Tank Flight Level Random Vibration Environment, All Three Axes

#### 4.5.3 FAME Instrument and Flight Vehicle

The flight level random vibration environment for the FAME system is given in Figure 4-12. The FAME system can either have the actual instrument or an instrument simulator. This specification applies to the test of the instrument alone.

If there is an engineering model available it is tested at a level +6 dB higher than flight level for a duration of 2 minutes. The subsequent flight unit is then tested at +3 dB higher than flight level for a duration of 1 minute.

If there is only a flight unit it is tested at +3 dB higher than flight level for a duration of 2 minutes.



##### Flight Level Environment

1.2 Grms Overall

Frequency (Hz)	G <sup>2</sup> /Hz
20	0.0020
100	0.0100
200	0.0100

All 3 Axes

##### Test Level

	Margin Above Flight Level (dB)	Duration (Minutes)
--	-----------------------------------	-----------------------

Engineering Model (Qualification Level)	6	2
Flight Spacecraft (Protoflight Acceptance)	3	1

Note: The Spectrum will be tailored to keep  
primary structural responses below  
Design Limit Load X 1.05

**Figure 4-12. FAME System Flight Level Random Vibration Environment**



## 4.6 Acoustic Design Limit Loads

### 4.6.1 Instrument and Flight Vehicle

The spacecraft acoustic environment is a function of the launch vehicle configuration, the fairing, and the fairing 3.0-inch thick acoustic blankets. The spacecraft shall survive protoflight acoustic loads that are +3 dB above flight levels (Figure 4-13) applied for 1 minute.

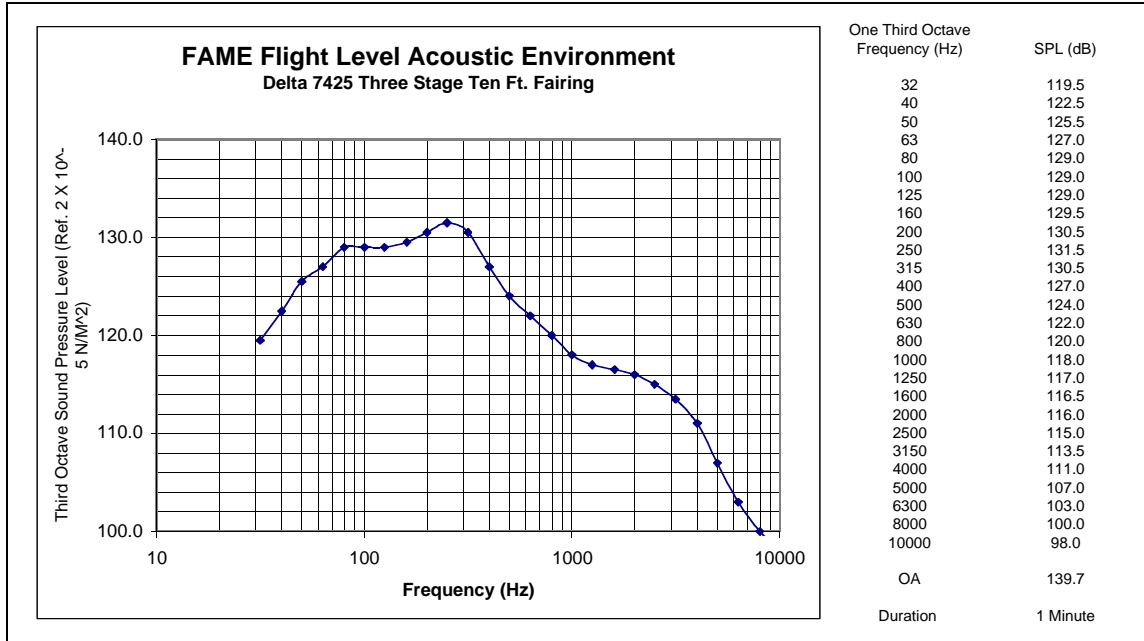
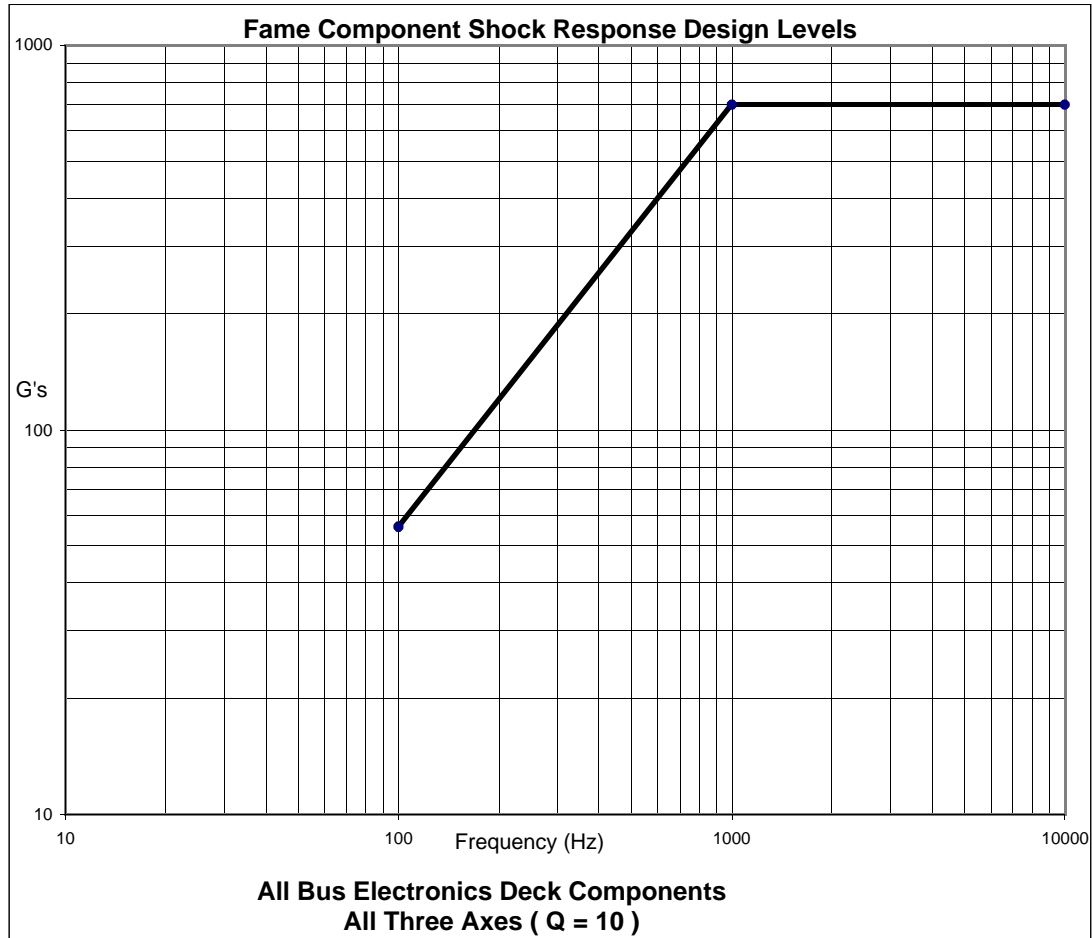


Figure 4-13. Predicted Delta II Flight Level Acoustic Environment

## 4.7 Shock

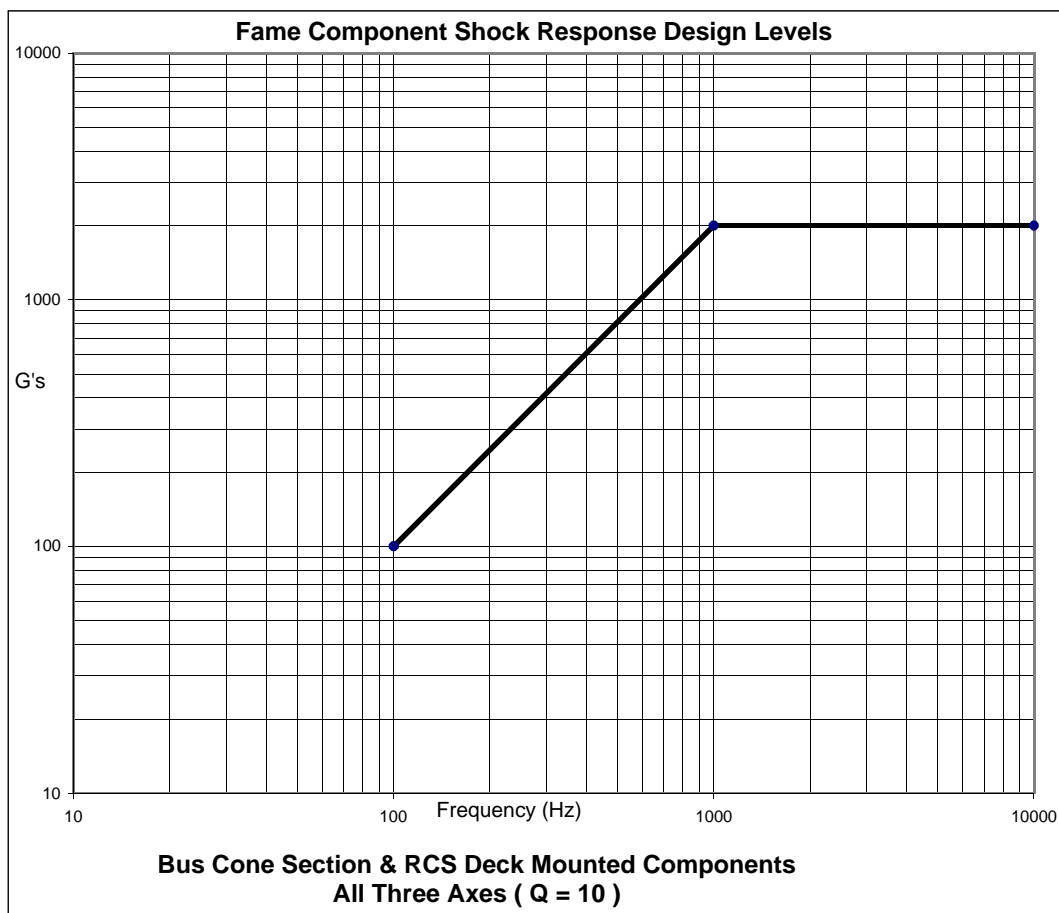
### 4.7.1 Component

The maximum shock environment at the spacecraft interface occurs during spacecraft separation from the flight vehicle and is a function of the spacecraft separation system configuration.



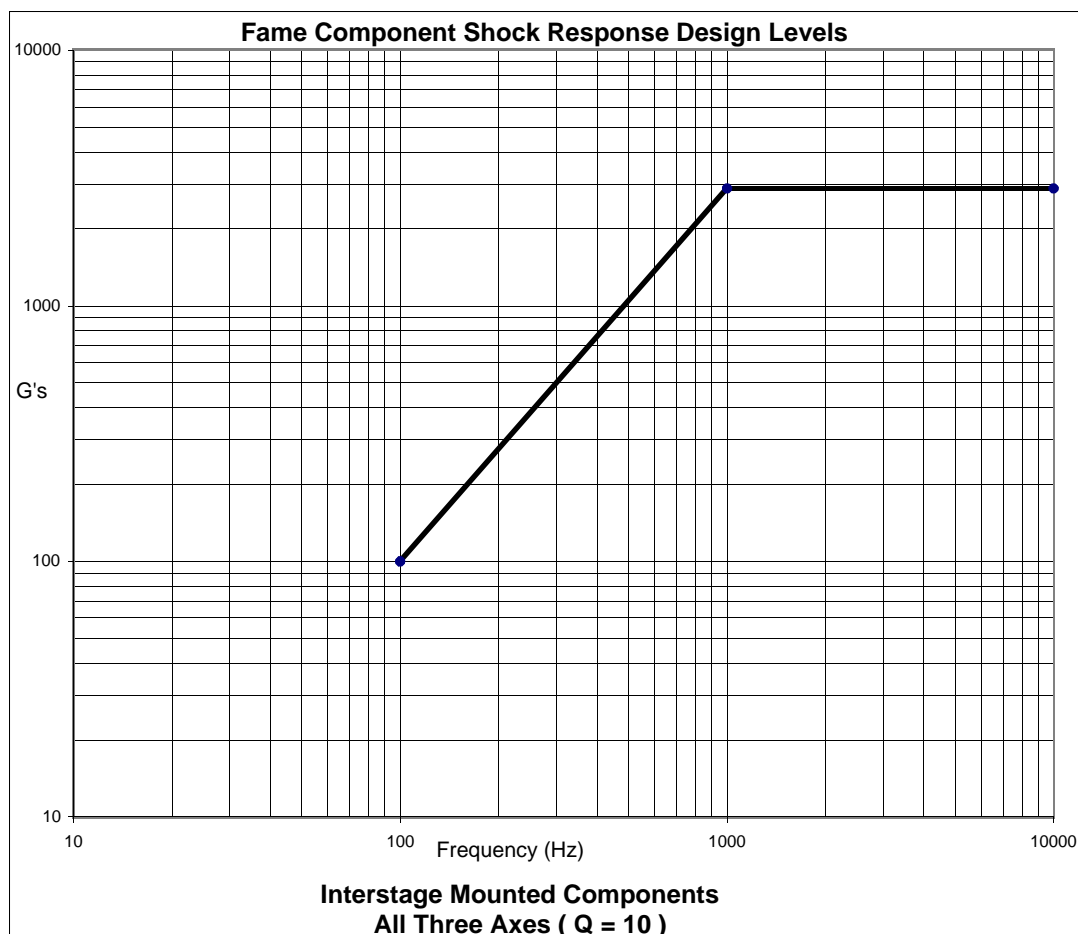
Design Environment Shock Response Spectrum Levels		Test Levels	
Frequency (Hz)	G's	Flight	1 Shock per Axis
100	56	Protoflight	2 Shocks per Axis
1000	700	Qualification	3 Shocks per Axis
10000	700		

**Figure 4-14. FAME Bus Mounted Components On Electronics Deck Shock Response Design Levels, All Three Axes (Q=10)**



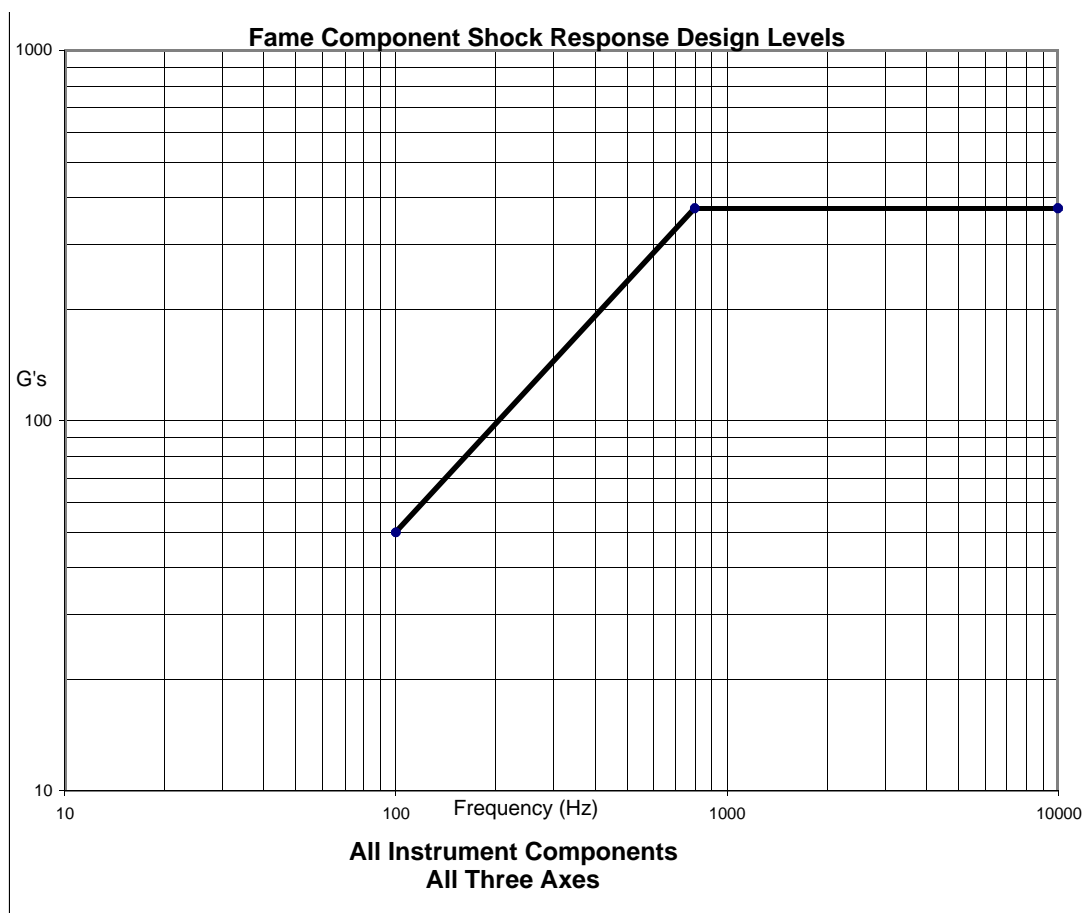
Design Environment Shock Response Spectrum Levels		Test Levels	
Frequency (Hz)	G's	Flight	1 Shock per Axis
100	100	Protoflight	2 Shocks per Axis
1000	2000	Qualification	3 Shocks per Axis
10000	2000		

**Figure 4-15. FAME Bus Mounted Components on Thrust Cone and RCS Deck Shock Response Design Levels, All Three Axes (Q=10)**



Design Environment Shock Response Spectrum Levels		Test Levels	
Frequency (Hz)	G's	Flight	1 Shock per Axis
100	100	Protoflight	2 Shocks per Axis
1000	2870	Qualification	3 Shocks per Axis
10000	2870		

**Figure 4-16. FAME Interstage Mounted Components Shock Response Design Levels, All Three Axes (Q=10)**

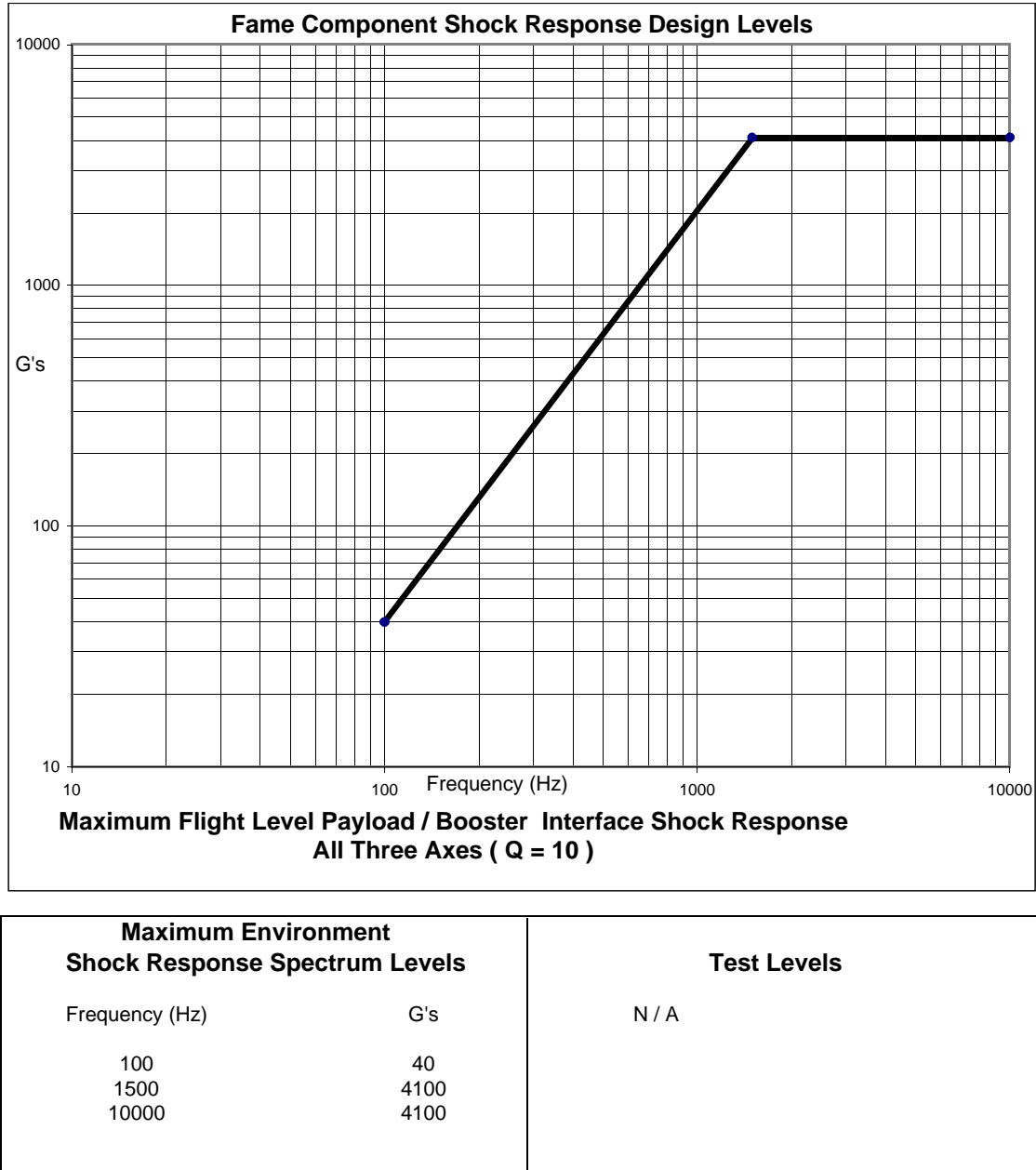


Design Environment Shock Response Spectrum Levels		Test Levels	
Frequency (Hz)	G's	Flight	1 Shock per Axis
100	50	Protoflight	2 Shocks per Axis
800	375	Qualification	3 Shocks per Axis
10000	375		
<b>Q = 10</b>			

**Figure 4-17. FAME Instrument Component Shock Response Design Levels, All Three Axes (Q=10)**

#### 4.7.2 System and Subsystem

The maximum shock environment at the PAF/spacecraft interface occurs during spacecraft separation from the launch vehicle and is a function of the PAF/spacecraft separation system configuration. The maximum shock response for a 3712 PAF is given in Figure 4-18.



**Figure 4-18. Maximum Flight-Level Payload Interface Shock Response Spectrum for 3712 PAF**

## 5. ANALYSIS REQUIREMENTS

### 5.1 Analysis Criteria

Analysis for the FAME spacecraft will consist of a combination of hand and finite element analysis for both stresses and deformation. Finite element models of the spacecraft and components will be used for modal analysis of the spacecraft and also to generate internal forces for detailed stress analysis. Load cases are generated from quasi-static flight and testing environments. Those load cases are expected to envelope results from coupled load analysis.

### 5.2 Responsibilities

It is the responsibility of the structural analysis team to assure that the structural and mechanical systems of the FAME program meet the structural requirements as specified. In order to accomplish this, the structural analysis team will:

- a) Conduct the following analyses:
  - Initial loads analysis
  - Coupled loads analysis model and LTM preparation
  - Transportation and handling loads analysis
  - Test loads analysis - pre and post
  - Detailed stress analysis including margins of safety
  - Normal modes analysis
  - Deflection and deformation analysis
  - Flight dynamics analysis - jitter, etc.
  - Thermal distortion
  - Manufacturing support
- b) Develop necessary finite element models of the spacecraft and payload adapter fitting.
- c) Provide support and direction to the design group in selection of load paths, part sizes, and materials.
- d) Provide support in the development of structural and environmental test plans and subsequent testing.
- e) Provide pretest structural and environmental test predictions.
- f) Review and correlate the test data with mathematical models.
- g) Develop the final verified analytical models of the flight system.
- h) Review analyses of vendor supplied components.

### 5.3 Analysis Approach

Analysis for the FAME program will consist of a combination of hand and MSC/NASTRAN finite element analysis. Applied loads will be the program defined design limit loads, component loads due to the mass-acceleration curve, test loads including random vibration for components, and transport and handling loads. The flight vehicle and spacecraft will be designed to the design masses and Cg locations. Standard detailed stress analysis techniques will be used to calculate component allowable loads, stresses and margins of safety. Element and grid point forces used for detailed stress analysis as well as deflections and modal frequencies are obtained from correlated finite element models of the spacecraft. Thermal loads and distortions will be addressed on an as-needed basis.

### 5.4 Applied Loads

The Delta II provides preliminary quasi-static loads. These loads will provide the basis for developing the Design Limit Loads (DLL) for the spacecraft. The flight vehicle, spacecraft, including both primary structures and PAF, and selected subsystems such as the instrument will use the DLL previously defined in paragraph 4.4.2 for analysis. If there are any test, handling or transport loads not enveloped by the DLL they will also be analyzed.

Components will be analyzed to meet all test loads, including random vibration. The MAC will be applied to all components and to their means of attachment. Bracketry will be analyzed for the MAC and any test loads. The MAC curve and its application are discussed in paragraph 4.4.1.

## 5.5 Factors of Safety (FOS)

### 5.5.1 Structural Factors of Safety

	With Test:	No Test (see Note):
Yield	1.10	1.60
Ultimate	1.40	2.00
Composite Ultimate	1.50	N/A
Local Buckling	1.25	1.60
Overall Stability	1.40	2.00
Fitting Factor (single point failure)	1.15	1.15
Gapping	1.15	1.15
Fatigue (lifetimes)	4.00	4.00
Bonded Joint Adhesive, Yield	1.15	N/A
Bonded Joint Adhesive, Ultimate	1.50	N/A
Mechanical Test Level	1.05	N/A
Composite Proof Test Level	1.10	N/A

Note: Applies to metallic components and secondary structures only

### 5.5.2 Pressurized Components Factors of Safety

Pressurized components including vessels, lines and fittings shall be designed and analyzed in accordance with MIL-STD-1522A, Approach A. Analysis shall verify that there will be no yielding at proof levels and the strength will be adequate to reach the tabulated burst levels. Minimum material thicknesses and allowable strengths with appropriate temperature reductions will be used in all calculations. Proof and burst factors are based on the Maximum Expected Operating Pressure (MEOP) of the components.

	Proof Factor	Burst Factor (BF)
Pressure Vessels (BF<2:1)	$(1+BF)/2$ (see note)	1.5
Pressure Vessels (BF>2:1)	1.50	2.00
Lines and Fittings < 1.5 inch diameter	1.50	4.00
Lines and Fittings > 1.5 inch diameter	1.50	2.50
Valves, Filters, etc.	1.50	2.50

Note: Or proof level determined by Fracture Mechanics Safe-Life Analysis (Minimum 1.25)

### 5.5.3 Ground Support Equipment

Design and analysis shall verify that the equipment is adequate to meet the following FOS. An additional Fitting Factor of 1.15 shall be applied to single point failure joints.

	Proof Test Factor	Yield FOS	Ultimate FOS
Handling Equipment	2.00	3.00	5.00
Lifting	2.00	3.00	5.00
Work Stands	2.00	3.00	4.00
Other	1.50	2.00	3.00

## 5.6 Margins of Safety

All analysis performed shall demonstrate a margin of safety of +0.00 or higher for yield and ultimate load conditions. Margin calculations shall be generated using standard equations or appropriate interaction equations.



$$MS_{yld} = \frac{(\text{Allowable yield load or stress})}{(\text{Limit load or stress} \times FS_{yld})} - 1$$

$$MS_{ult} = \frac{(\text{Allowable ultimate load or stress})}{(\text{Limit load or stress} \times FS_{ult})} - 1$$

## 5.7 Mass Properties for Analyses

The mass properties, both weight and Cg locations, used for the analysis have been previously defined in this document. Discrepancies between the baseline mass properties and model mass properties shall be resolved by adding nonstructural mass to the model.

## 5.8 Finite Element Analyses

### 5.8.1 Models

Finite element models of the flight vehicle will be used to generate internal loads in the primary structure from the quasi-static design loads, obtain displacements of critical points, and predict modal frequencies and mode shapes. Offline finite element models of individual components may be used to obtain stresses and modal information when necessary.

### 5.8.2 Model Checkout

Models will be reviewed prior to use for correctness. Symmetry, correct mass properties, grounding, free-free and constrained modes will be evaluated as part of the checking process. Modal effective weight tables shall be used to identify primary modes.

### 5.8.3 Model Results

Results from the finite element model include internal loads and grid point forces for detailed stress analysis, modal frequencies, and mode shapes for correlation with modal testing, and nodal deflections.

## 5.9 Detailed Stress Analysis

### 5.9.1 Primary Structure

Primary structure is defined as all structural components that make up the primary load paths of the structural system. The primary structure for the FAME spacecraft consists of a composite thrust tube, a PAF, deck angles, and a skeleton frame of longerons and instrument support panels. The thrust tube provides structural mounts for the longerons and deck angles. Detailed stress analysis of the primary structure shall include strength, local stability, bearing strength, and any other analysis deemed necessary depending on the function of the structural component.

### 5.9.2 Secondary Structure

Secondary structure is defined as all structural components that are not part of the primary load path. In general, secondary structural components include mounting brackets, non-structural closure panels, equipment support decks and other equipment support structures. The solar array/sun shield is considered a secondary structure. Secondary structures may be designed using the MAC or quasi-static loads depending on the nature of the structure. Secondary structures shall be analyzed for the effects of test and flight vibro-acoustic test requirements in addition to the quasi-static loads. The FAME System Test Plan, SSD-TP-FM001, provides the vibro-acoustic test requirements. Allowable loads for the structural components shall be calculated using standard analysis techniques.

### 5.9.3 Buckling

Global stability of the flight system shall be analyzed as an ultimate failure mode. Local stability (crippling, component buckling) shall be analyzed in the detail stress analysis. Panel buckling shall be addressed with its own factor of safety.

#### 5.9.4 Fasteners and Bolted Joints

Fasteners and bolted joints shall be analyzed for fastener strength, bearing on components, and strength of fittings. Fastener preload, without an FOS, shall be included with the applied load. Gapping margins where necessary shall also be calculated. The strength of the nuts shall be evaluated in tension-type bolted joints.

NRL shall assume the responsibility for the analysis of the attachment of the instrument flexures to the bus.

#### 5.9.5 Stress Concentrations and Fatigue Analysis

Stress concentrations may produce stresses above the material's elastic limit, which can lead to fatigue failure after only a few number of load cycles. For this reason, fatigue analysis shall be performed whenever significant stress concentrations exist. For materials, which exhibit brittle fracture, stress concentrations shall be applied in static analysis. Stress concentrations shall not exist in the primary structure but may be permitted for secondary structure if static and fatigue analysis demonstrate sufficient margin and life.

### 5.10 Load Cycle Program

Coupled Loads Analysis (CLA) shall be performed to assure that the DLL factors are adequate and that positive margins of safety are maintained for flight events. The DLL factors shall be modified upward if the CLA demonstrates that generally higher loads are occurring for flight events. If just local areas show loading above the DLL factors those areas shall have their margins calculated with the results of the CLA.

CLA shall be performed by the Launch Vehicle Dynamics group. The Craig-Bampton model of FAME shall be coupled to the structural model of the launch vehicle. Interfaces shall be as described below. Forcing functions are applied for various flight events. The direct results are then multiplied by the FAME-supplied transformation matrices (size TBD) to obtain local stresses, loads, accelerations, and displacements of concern.

The instrument manufacturer shall provide NRL with an instrument Craig-Bampton model for each loads cycle. The Instrument C-B model shall be coupled to the FAME bus model. Interfaces shall be as described below. LTMs (size TBD) shall also be provided for the recovery of instrument results.

A Preliminary Design Load Cycle (PDLC), Final Design Loads Cycle (FDLC), and a Verification Load Cycle (VLC) shall be performed and scheduled to coincide with the Preliminary Design Review (PDR), Critical Design Review (CDR), and the Verification Readiness Review (VRR). Additional load cycles may be performed as required.

PDLC and FDLC use FAME (with integrated instrument) finite element models at their current state of maturity. There is no effort made to correlate the model since test data are not available at the time.

The VLC model shall be correlated to actual structural test data. The model shall be modified to match primary frequencies and mode shapes.

#### 5.10.1 Model Uncertainty Factors

Model Uncertainty Factors (MUF) are applied to the initial results of the coupled loads analysis to account for the lack of model maturity. The MUFs to be used for this program are:

PDLC = 1.50

FDLC = 1.25

VLC = 1.00 (Assuming adequate correlation)

The lack of adequate correlation (frequency and mode shape) in any component of the VLC model shall require a MUF (1.25 TBR) on the results for that component.

#### 5.10.2 Verification Loads Cycle Models

The models used for the VLC, including the Instrument, shall be test correlated using the following process and criteria.

### 5.10.2.1 Modal Test

A modal test shall be performed to obtain the necessary mode shape and frequency information to perform the model correlation. The information shall be gathered for critical target modes.

It is anticipated that the instrument shall be tested alone and the entire flight vehicle shall be tested utilizing Instrument and AKM mass simulators.

### 5.10.2.2 Target Modes

The target modes shall be those that are considered critical because they contain large percentages of the modal effective weight, contribute to the forces in critically loaded components, or show high response to launch vehicle flight excitations.

### 5.10.2.3 Correlation Criteria - Frequency

The target test and analytical modes shall be compared to verify frequency similarity. Target mode analytical frequencies shall match test measured frequencies within 5%.

### 5.10.2.4 Correlation Criteria – Modal Shape Cross Orthogonality

Mode shape cross orthogonality is defined as:

$$[XORTHO]=[Φ_E]^T [M_A] [Φ_A]$$

Where:

$[Φ_E]^T$  = Experimental Modes (transpose)

$[M_A]$  = Analytical Mass Matrix reduced to measurement points

$[Φ_A]$  = Analytical Modes

The correlation goal shall be 0.90 terms or greater on the diagonal and 0.15 or less off diagonal.

### 5.10.3 Instrument to FAME Interface

The instrument to FAME interface for coupled loads models shall be defined as:

Number of points TBD

Coordinate system TBD

Coordinates TBD

### 5.10.4 FAME to LV Interface

The FAME to LV interface for coupled loads models shall be defined as:

Number of points TBD (one point collapsed at interface or multiple bolt locations or ?)

Coordinate system TBD

Coordinates TBD

## 5.11 Deliverables and Documentation

Documents and Deliverables	Purpose and Format	From	To	Due Date
Instrument PDLC Model, LTMs, and Documents	Instrument Craig-Bampton PDLC Model and LTMs. LTM size TBD	LMMS	NRL	System PDR
SC PDLC Model, LTMs, and Documents	SC Craig-Bampton PDLC Model and LTMs. LTM size TBD	NRL	LV	System PDR
FDLC Results	LTM results for all flight load cases	LV	NRL	System PDR
Instrument PDLC Model, LTMs, and Documents	Instrument Craig-Bampton PDLC Model and LTMs. LTM size TBD	LMMS	NRL	System CDR
SC PDLC Model, LTMs, and Documents	SC Craig-Bampton PDLC Model and LTMs. LTM size TBD	NRL	LV	System CDR
FDLC Results	LTM results for all flight load cases	LV	NRL	System CDR
Instrument Correlation Report	Summarizes correlation of model to modal test data/final summary of correlation. Identifies target modes. Includes frequency comparison. Includes cross ortho check.	LMMS	NRL	VLC
Instrument Modal Test Report	Describes the modal test and presents results.	LMMS	NRL	VLC
Instrument VLC Model, LTMs, and Documents	Instrument Craig-Bampton VLC Model correlated to modal test. Interfaces, coordinate systems, mass properties, modes, and LTMs are described in the report. LTM size TBD	LMMS	NRL	VLC
SC Modal Test Report	Describes the modal test and presents results.	NRL	NRL	VLC
Flight Vehicle Correlation Report	Summarizes correlation of model to modal test data/final summary of correlation. Identifies target modes. Includes frequency comparison. Includes cross ortho check.	NRL	NRL	VLC
Flight Vehicle VLC Model, LTMs and Documentation	Flight Vehicle Craig-Bampton VLC Model correlated to modal test. Interfaces, coordinate systems, mass properties, modes, and LTMs are described in the report. LTM size TBD	NRL	LV	VLC
VLC Results	LTM results for all flight load cases.	LV	NRL	VLC
Instrument Stress Report	Summarizes analysis results and margin of safety table with supporting analysis. Design loads verified by VLC. Includes model description.	LMMS	NRL	TBD
Instrument Structural Test Report	Document results of static and dynamic testing of primary structure. Describes test objectives, areas tested, configuration, test approach and test results.	LMMS	NRL	TBD
Flight Vehicle Stress Report	Summarizes analysis results and margin of safety table with supporting analysis. Design loads verified by VLC. Includes model description.	NRL	NRL	TBD
Flight Vehicle Structural Test Report	Document results of static and dynamic testing of primary structure. Describes test objectives, areas tested, configuration, test approach and test results.	NRL	NRL	TBD